Interstellar Probe Mission/System Concept

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Abstract

The first mission in NASA's Interstellar Program, Interstellar Probe (ISP), will be the first to probe the interstellar medium with a complete set of scientific instruments designed for such exploration and is expected to be a precursor and a significant testbed for technologies being developed for eventual travel to the nearest star. The interface between our solar system and galaxy defines the crossover into the interstellar medium and is the minimum target distance for ISP, thought to be beyond 125 AU. A mission requirement, therefore, is to reach 200 AU in 15 years or less with a scientifically capable payload package. Time and distance are key design requirements, and advanced propulsion technology is a key enabler of the Interstellar Probe mission. Another key mission goal is to launch in the mid-term (2010) time period, which establishes associated advanced technology goals of readiness by about 2007. Solar sail propulsion has been baselined for the mission design concept. Key trades are sail technology development requirements as a function of trip time to 200 AU and the payload mass that can be delivered and operated at that distance. This paper provides strawman payload and measurement requirements, technology and mission trade information, and a baseline system design, including a configuration concept. Alternate technology options are also described.

I. INTRODUCTION. Technology advances in the space exploration arena appear to be accelerating at a rate difficult to have been predicted only a short time ago. Our vision of missions that might be has often fallen behind what could be. This paper suggests that the first stage of interstellar exploration is ready to begin, as suggested and outlined in Wallace 1999. NASA's Interstellar Program was begun in the Spring of 1999 after a year of advanced mission and program planning activities, reported previously in a paper delivered at the 1999 IEEE Aerospace Conference. The Interstellar Probe (ISP) mission, the first to probe the interstellar medium (ISM) with a complete set of scientific instruments designed for such exploration, is expected to be a precursor and a significant testbed for technologies being developed for eventual travel to the nearest star.

Summarized in this paper is the material developed over a five-month period by a JPL team of mission/system designers and advanced technology developers for presentation to NASA Headquarters in the summer of 1999. The rationale for the first mission is based on being able to explore previously unexplored regions of space with innovative technology that has recently come onto the scene. The focus is an end-to-end mission/system design that is integrated and based on technology predicted to be in place by about 2007. Selection of solar sail technology for the baseline design was made on the basis of available information regarding readiness and performance (Garner, Diedrich, and Liepold 1990). Other propulsion system candidates continue to be considered as options to this baseline.

The baseline design is described in an evolutionary manner, with science requirements defined first, mission requirements and baseline architecture next, and, finally, the resulting flight system design. New technology is the basis for the design; this driving factor is brought into the design at all levels. In particular, mission performance as a function of solar sail technology capability is an important trade in the mission architecture and analysis described in Section III, Mission Concept Architecture and Design.

II. Science Objectives, Payload, and Measurement Requirements. Interstellar Probe's expedition to 200 AU and beyond provides the first comprehensive set of measurements of plasma, neutrals, dust, magnetic fields, energetic particles, cosmic rays, and infrared emission from the outer solar system through the boundaries of the heliosphere and into the ISM (Mewaldt 1998; Liewer 2000). This voyage enables the achievement of scientific goals dating back to the early 1990s (Holzer, Mewaldt, and Neugenbauer 1990). Specifically, the principal scientific objectives of the Interstellar Probe mission are to:

1. Explore the nature of the interstellar medium and its implications for the origin and evolution of matter in our galaxy and the universe
2. Explore the influence of the interstellar medium on the solar system, its dynamics, and its evolution
3. Explore the impact of the solar system on the interstellar medium as an example of the interaction of a stellar system with its environment
4. Explore the outer solar system for clues to its origin and to the nature of other planetary systems

To achieve these broad interdisciplinary objectives, the strawman scientific payload includes an advanced set of miniaturized, low-power instruments specifically designed to make comprehensive, in-situ studies of the plasma, energetic particles, fields, and dust in the outer heliosphere and nearby ISM. The instruments that comprise the scientific payload, along with individual measurement requirements, are compiled in Table 1.

Table 1: Strawman Instrument Payload and Measurement Requirements

Instrument	Measurement Requirement
Magnetometer	~ 0.001 nT in 3 axes
Plasma and Radio Wave Sensor	E field, 5 Hz to 5 kHz
Solar Wind/Interstellar Plasma/Electrons Spectrometer	H, He, 10 eV to 30 keV/q; Electrons: 1 eV to 2 keV
Pickup and Interstellar Ion Composition Spectrometer	Ions of $1 \le Z \le 26$ from 10 eV to 30 keV/q
Interstellar Neutral Atom Spectrometer	${}_1H$, ${}_2H$, ${}_3He$, ${}_4He$, C, O; E/q, 10–800 eV
Suprathermal Ions/Electron Sensor	E/q of ions, $2 \le Z \le 26$; Ions: 20 keV/q to 2 MeV/q e^-: 10 keV to 1 MeV
Cosmic Ray H, He, Electrons, Positron, γ-Ray Burst Instrument	H, He: 3 to 130 MeV; e^-: 10 keV to 1 MeV e+, gammas: 0.2–10 MeV
Anomalous and Galactic Cosmic Ray Composition Spectrometer	$3 \le Z \le 30$, $6 \le M \le 28$; $1 \le E \le 300$ MeV/n
Dust Composition Instrument	Mass and composition for $M > 10^{-13}$ kg
Infrared Instrument	~ 2 to ~ 150 _m
Energetic Neutral Atom (ENA) Imager	Energetic neutrals in the energy range 0.3 to 7 keV
Ultraviolet Photometer	Lyman alpha flux

III. Mission Concept Architecture and Design

Mission Requirements

To fulfill the science objectives of the mission, key requirements for the mission and system design were developed. These governing requirements stem not only from the science objectives, but also from programmatic and technology considerations. In particular, technology maturity, development programs, and agency (NASA) theme timelines were considered. The primary requirements, as applicable to the ISP mission and system design (not all-inclusive), are summarized in Table 2.

The reference design is the result of mission and system trades within technology and performance design spaces. The mission and system concept, as developed at JPL and discussed in this paper, met all science requirements. The key trades affecting the baseline mission follow.

Key Mission Trades

To establish the baseline mission design, an assessment of key mission parameters was performed. Principal to the mission design were trades to understand how sail areal density, perihelion distance, and sail jettison range affect flight time and delivered flight system mass.

Table 2: Key Mission and System Requirements

Category	Requirement
Technology Cutoff Date	2007
Project Start	2007
Launch Date	2010
Launch Vehicle	Delta IV class vehicle or smaller
Trajectory Design	Flight path: Target for 'nose' of heliopause Sail jettison: ≤ 50 AU Flight time: 200 AU in ≤ 15 yrs Goal: 10 yrs Range goal: 400 AU Perihelion range: 0.25 AU Launch C^3: 0 km^2/s^2
Flight System	Design Lifetime: 15 yrs Consumables: Size for 30 yrs (400 AU range goal) Payload accommodations: Number of instruments: 12 Mass: 25 kg Power: 20 W Observation mode: Spinning platform Science data: F/S DPUs for processing Provide science storage Science acquisition: Continuous at > 50 AU Science downlink: 25 bps at 200 AU
Solar Sail	Sail areal density: 1 g/m^2 Implementation: Spinning Jettison deployment canister/ mechanisms prior to acceleration
Data Return Strategy	DSN coverage: ≥ 1 pass/wk Selected high-rate activity periods

Sail areal density (sail subsystem mass/sail area, σ_s) is a function of material properties, design implementation (includes control scheme), and technology readiness. Downselection to a rotationally stabilized sail implementation occurred after consideration of other sail concepts (Wallace 1999). Central to this selection is the estimate of a 1 g/m^2 sail areal density being achievable for this design concept, consistent with a technology readiness date of 2007.

The radius of perihelion can be varied to allow for faster flight times to the target. The selection of the radius of perihelion for the reference mission was a function of the sail's capability to maintain performance for a given solar flyby environment.

Figure 1 illustrates the trade among flight system mass delivered, sail size, and different radii of perihelion for a fixed sail areal density. This design space, coupled with the projected technology readiness of the sail materials and the required flight system mass, allows the selection

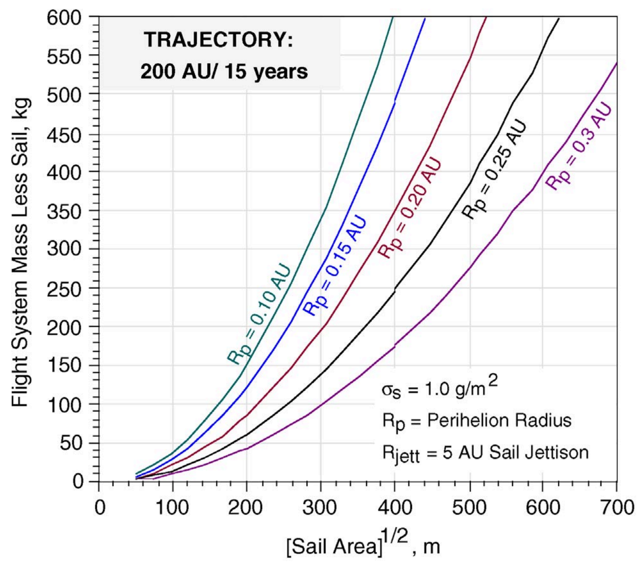

Figure 1. Sail performance, radius of perihelion trade space

of a reasonable solar flyby distance (0.25 AU) that minimizes risk and design complexity.

The science requirement to begin data acquisition, free of sail interactions, by 50 AU, led to trades in sail jettison range. Figure 2 addresses flight system mass and sail size as a function of sail jettison range and

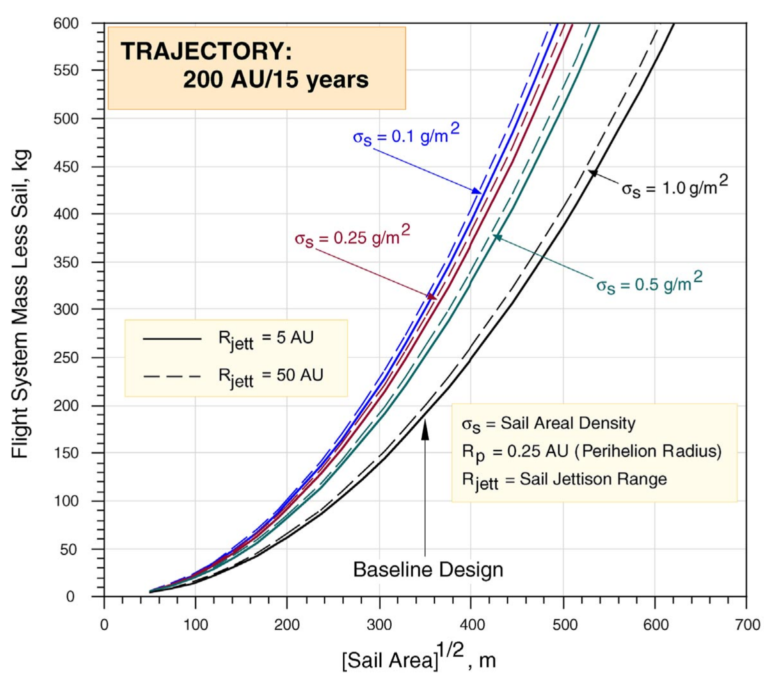

Figure 2. Sail performance, sail jettison range trade space with areal density

sail areal density. As can be seen, there is very little difference in the flight system mass delivered (≈ 7 kg) between sail jettison ranges of 5 and 50 AU (for the baseline mission with a fixed sail area).

Figure 3 presents the general case for flight time trades relative to delivered mass, sail size, and areal density. The desired mission parameters are defined by the cross-hatched box and were used, along with the trades in jettison range and perihelion distance, as a starting input for the baseline mission assessment.

Trajectory Design

The solar sail trajectory design described below is based on the analysis described in Sauer 1999. A solar sail is used to increase the energy of the flight system's heliocentric orbit in order to achieve escape in the direction of the nose of the heliopause. The best location for increasing the orbit's energy is, in general, at perihelion. A sail's ΔV is, however, the sum of a continuously applied low thrust that varies in magnitude with heliocentric range. The incremental energy change is proportional to the velocity at which the ΔV is applied and is maximized when the applied ΔV is parallel to the velocity vector. Since the acceleration performance of a solar sail increases as the sail gets closer to the Sun (i.e., photon pressure increases), as close a solar flyby as possible is necessary to maximize the thrust performance of the sail, thereby maximizing delivered mass and minimizing flight time.

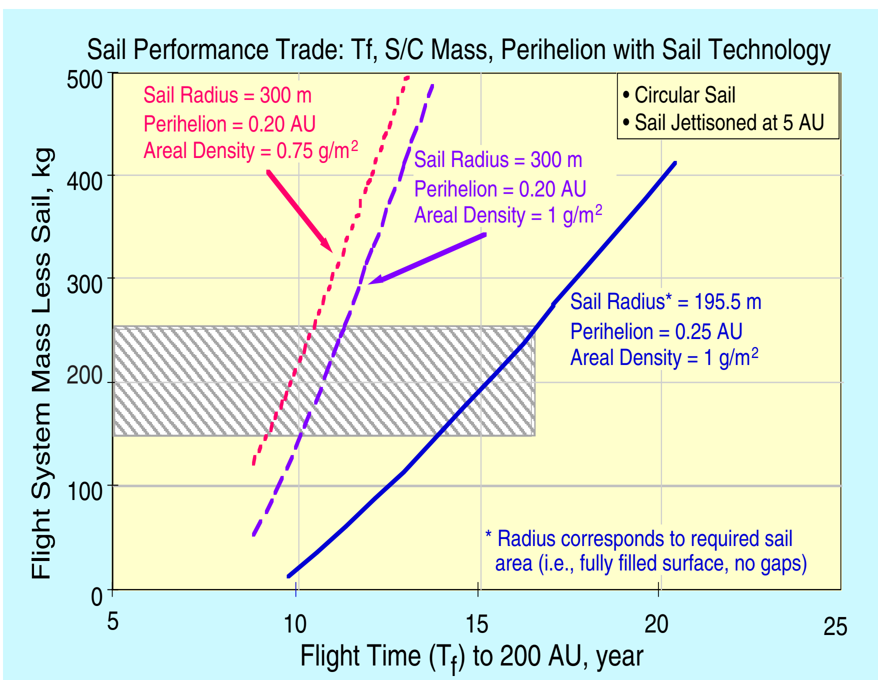

Figure 3. Flight time to 200 AU mission trade space

To optimize the energy gain of the trajectory, the sail is first used to reshape the flight path from Earth orbit into a close high-speed solar flyby, such that the high photon pressure at perihelion imparts a large acceleration to the flight system. The close solar flyby increases the effectiveness of the sail so much so that the heliocentric energy of the flight system can be increased from negative to positive (with the necessary departure velocity) in just one perihelion passage.

The baseline trajectory was optimized using a model of the sail as a flat perfect reflector so that the thrust from the pressure of the sunlight is normal to the sail. With this assumption, the flight time is a function of only the characteristic acceleration, a_c, and the perihelion radius.

For this study, the performance of the flight system sail was derated to 85% of that of an "ideal" sail. This derating allows for such physical properties of the sail as less than 100% reflectivity, photon absorption and re-radiation, and figure (shape) errors in the local and general surface of the sail. The effective thrust available as a function of the Sun incidence angle (angle of the Sun relative to the sail normal) varies as the square of the cosine of the incidence angle for an ideal sail. A more accurate (realistic) simulation of sail performance requires further development of sail models and algorithms. For the given maturity of the flight system design, however, the derating to 85% is considered a conservative approach to sail performance estimation.

Using these models and assuming a minimum perihelion distance of 0.25 AU, several flight path options were examined to determine the required performance of the sail (i.e., a_c). Trajectories with 10- to 30-year flight times were investigated within the trade space. The reference trajectory selected is characterized by a 15-year flight time to 200 AU.

This trajectory requires a "total sail loading" (not to be confused with "sail areal density") of < 2.55 g/m^2 to achieve the required characteristic acceleration of 3.039 mm/s^2.

Figure 4 illustrates the reference trajectory, including sail orientation along the flight path. Heliocentric and geocentric spacecraft range as a function of time is captured in Figure 5. Key angles of the spacecraft relative to the Earth and Sun are given in Figure 6. Note that the Earth is sometimes nearly face-on to the sail and sometimes edge-on to the sail, presenting challenges to the telecommunications system design within the inner solar system.

Mission Profile

The primary events and phases of the baselined 200 AU-in-15-year mission are shown in Figure 7. The solar sail is deployed within the first few days after launch. During the first 30 days of the mission, there is continuous DSN coverage using the 34-m subnet for tracking and spacecraft characterization. After this initial period, weekly DSN passes using the 34-m subnet are used for spacecraft tracking, health and safety

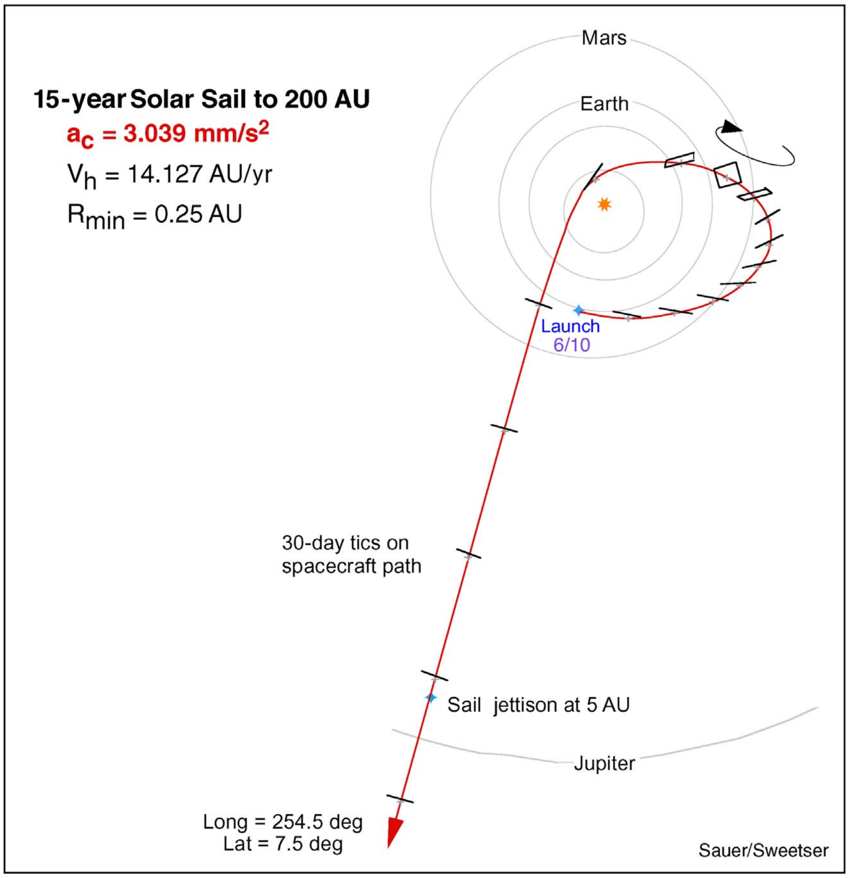

Figure 4. Baseline solar sail trajectory, 200 AU in 15 years

assessment, and uplink commanding through 10 AU (which occurs at Launch+19 months).

Perihelion is achieved approximately 11 months after launch (0.25 AU). The solar sail is jettisoned at 5 AU (Launch+15 months) and is followed by a spacecraft-sail separation maneuver. Between 5 and 10 AU, high-rate science data are collected (250 bps). Low-rate science (25 bps) begins at 10 AU and continues for the duration of the mission. Science data playback is accomplished using the DSN 70-m subnet, with one pass per week required until 150 AU (Launch+11.5 years), at which time two passes per week will be required.

IV. BASELINE FLIGHT SYSTEM DESCRIPTION. The flight system functionally consists of three major elements: the first is used to deploy the sail, the second is the sail itself and, in its final configuration, the third element consists of the spacecraft bus with all 12 instruments but without the sail, which is jettisoned at 5 AU.

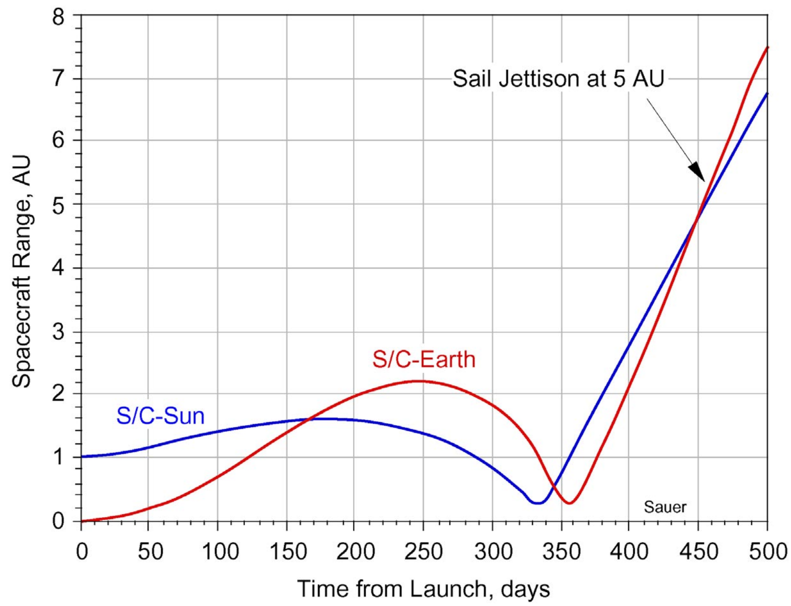

Figure 5. Baseline solar sail trajectory, flight system range vs. time

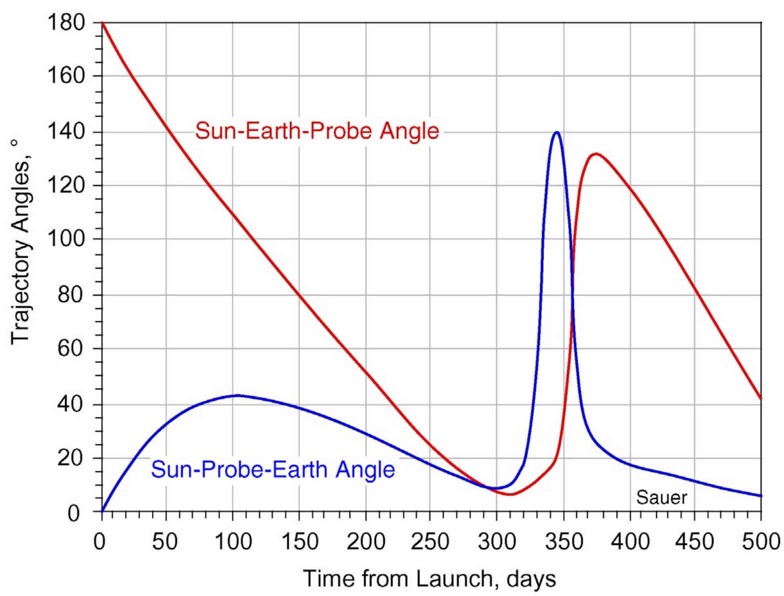

Figure 6. Baseline solar sail trajectory, characteristic angles vs. time

During launch, the sail is stowed inside a canister attached to both the spacecraft and the deployment module. After the launch event, the 400-m diameter spinning sail is deployed by extending three 10-m booms with cold gas systems and then rapidly rotating the whole structure. The deployment device is immediately jettisoned after sail deployment.

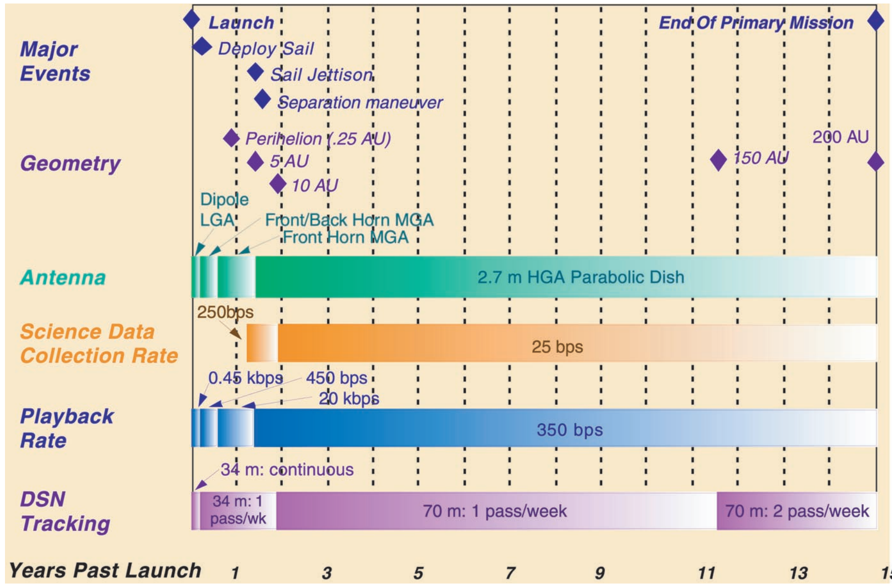

Figure 7. Reference mission profile

Solar photon pressure on the sail is then used to decrease the spacecraft velocity, such that the spacecraft swings into the inner solar system and around the Sun with a perihelion of 0.25 AU. After accelerating the spacecraft away from the Sun, the sail is jettisoned at 5 AU. This baseline design provides a cumulative ΔV of > 70 km/s, propelling the spacecraft to a distance of 200 AU from the Sun in less than 15 years (Wright 1992).

The spacecraft, exclusive of the sail and its deployment hardware, is dominated by its antenna. The 2.7-m rigid antenna functions as the main structure of the spacecraft, with at least 12 instruments arrayed along its rim. The Ka-band telecommunications subsystem, Reaction Control System (RCS), and Alkali Metal Thermal-to-Electric Converters (AMTECs) are located at the base of the antenna.

At launch, this bus is attached to the sail canister, which is connected to the sail deployment canister (mechanisms for unfurling the solar sail). Once the solar sail is deployed, the majority of deployment mechanisms are jettisoned to minimize the mass that must be accelerated.

While sailing, the movement of the sail is controlled by moving the spacecraft away from the center of mass of the spacecraft with respect to the sail's center of pressure. Movement is provided by translation along a rail located at the center of the sail.

In its final configuration, the spacecraft is in a very slow spin, with three 30-m-long deployed instrument booms. Due to the extreme length of the mission, as much as possible of the main spacecraft is block redundant. There is also a 30 percent contingency added to the mass and power tables to allow for expected growth in the system.

The power subsystem will provide and distribute power to the spacecraft, as defined by the mission power modes. The sizing of the power system was driven by a 30-year mission lifetime goal, science operating requirements, and the telecommunications subsystem. The study assumed that consumables would be sized for the 30-year mission. This resulted in a 316-W beginning-of-life system supplied by three next-generation Advanced Radioisotope Power Source (ARPS) units. To prevent interference with the highly sensitive instruments, the spacecraft structure is required to have magnetic characteristics within the 0.01-nT range and to combine multifunction elements as much as possible.

The RCS subsystem uses an advanced hot helium system with considerable redundancy to prevent leakage.

Table 3 represents the mass and power summary for the baseline flight system and an optional configuration considered (Option 2, representing a more aggressive sail design). Figure 8 shows the system block diagram for the referenced flight system concept.

An overview of each spacecraft engineering and instrument subsystem is given in the following sections. For each element, the following is addressed: driving requirements, baseline design description, mass, power, volume, and other options considered.

Baseline Design and Deployment Description

The spacecraft structure includes the following elements:

Launch vehicle interface
Solar sail cylinder and sail deployment module
Solar sail
Sun-shade
Spacecraft bus with instruments

<u>Launch vehicle interface</u>. For launch, the spacecraft is mounted atop the control rail and 1.5-m-long × 1.5-m-diameter sail cylinder, which, in turn, is mounted on top of the sail deployment module housing the sail deployment hardware and booms. The structural load path passes from the 1.2-m launch vehicle adapter straight through the sail deployment module and sail cylinder walls, picking up the ring at the base of the 2.7-m antenna (see Figure 9). After launch, the spacecraft is de-spun from the launch vehicle upper stage using a yo-yo device.

<u>Solar sail cylinder and sail deployment module</u>. Following de-spinning the spacecraft from the launch vehicle upper stage, three orthogonal booms are deployed from the 1.5 × 1.5-m deployment module. Thrusters are then fired, spinning up the spacecraft in a controlled fashion. Boom stowage cylinders are ~5% of the deployed length and about 20 cm in diameter. The deployed booms are ~10 m long, with thrusters on the end.

The sail is comprised of six pie-shaped triangles, folded into gores. At launch, the gores are wrapped around a 1.5-m-long × 1.5-m-diameter sail cylinder. As the spacecraft spins, the sail is released by sequentially

Table 3: Mass and Power Summary, Baseline and Option

	Dry Mass Fraction Solar Sail	Mass (kg) Baseline	Mass (kg) Option 2
Payload			
Instruments	17.2%	25.0	25.0
Payload Total	17.2%	25.0	25.0
Science Bus			
Attitude Control	2.5%	3.6	3.6
Command and Data	0.2%	0.3	0.3
Power (tri-AMTECs, 106 BOL ea.)	19.0%	27.5	27.5
Propulsion for RCS	6.0%	8.7	8.7
Structure, including Control	19.5%	28.3	28.3
Solar Sail Container/Adapter	1.5%	2.2	2.2
Cabling	4.0%	5.9	5.9
Telecomm (2.7-m HGA)	19.3%	28.1	27.6
Thermal (10 RHUs + Sun-shade)	10.8%	15.6	15.6
Science Bus Total		120.2	119.6
Science S/C Total *(Dry)*		145.2	144.6
Mass Power Contingency		43.6	43.4
Propellant/Pressurant for RCS	1.1%	2.1	2.1
Science Configuration *(Wet)*		190.8	190.1
Solar Sail		122.6	212.0
Sailing Configuration Total		313.4	402.1
Deployment Mechanisms Discarded			
L/V Structures		30.0	51.9
4 booms, 10-m long		30.0	51.9
Discarded Sail Structure		109.0	188.5
Launch Release Mechanisms		15.0	20.0
RCS (for Cold Gas System)		36.0	36.0
Total		220.0	348.2
Mass Contingency		66.0	104.5
Discarded Mechs./Structure Total		286.0	452.7
Launch Configuration Total		599.4	854.8
Launch Vehicle Capability		719.3	1200.0
Launch Vehicle Margin		119.9	345.2

releasing restraining devices that allow the gores to unfurl slowly and in a controlled manner to prevent snagging and collisions.

Tethers holding the center of each gore to the sail cylinder are then played out to form a "wheel rim" that is 410 meters in diameter. Each gore segment is unfurled into a triangle by pulling tethers, which connect each sail segment tip to the sail cylinder. Cartoons demonstrating the sail deployment sequence are shown in Figure 10 (Melnikov 1998).

After sail deployment, the sail deployment module, including the booms, spin-up assembly, and the launch vehicle interface structure are jettisoned from the spacecraft.

Solar sail. The baselined solar sail is 410 meters in diameter with an 11-m-wide central opening. The spacecraft module and its associated

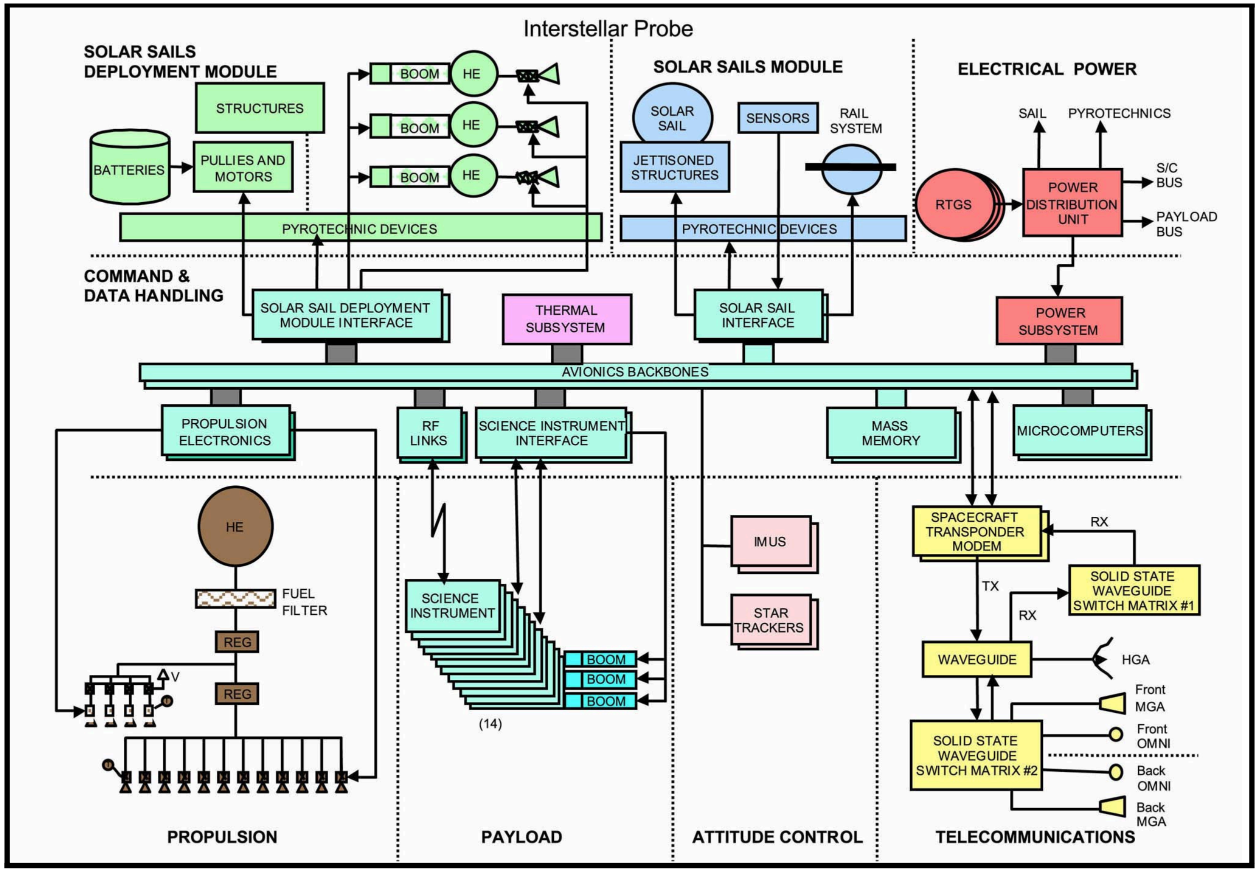

Figure 8. Flight system block diagram

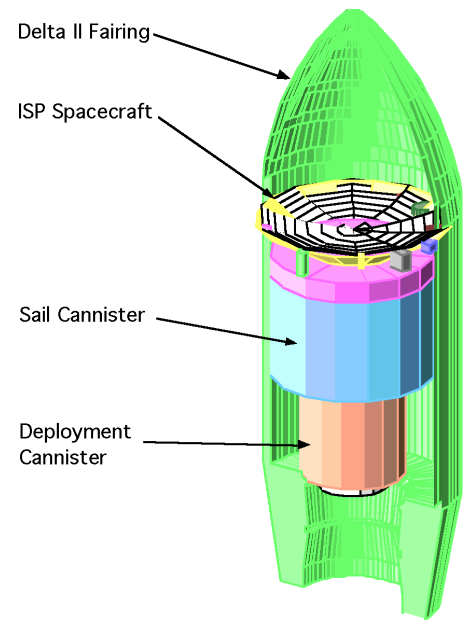

Figure 9. Spacecraft in launch configuration

Sun-shade are centered in the 11-m-diameter central aperture of the sail (see Figure 11 for a central view of the sail). Attitude control and thrust vector pointing of the sail are provided by moving the spacecraft mass relative to the system center of pressure. This is accomplished by moving the spacecraft on a short rail. The spacecraft can move up and down the

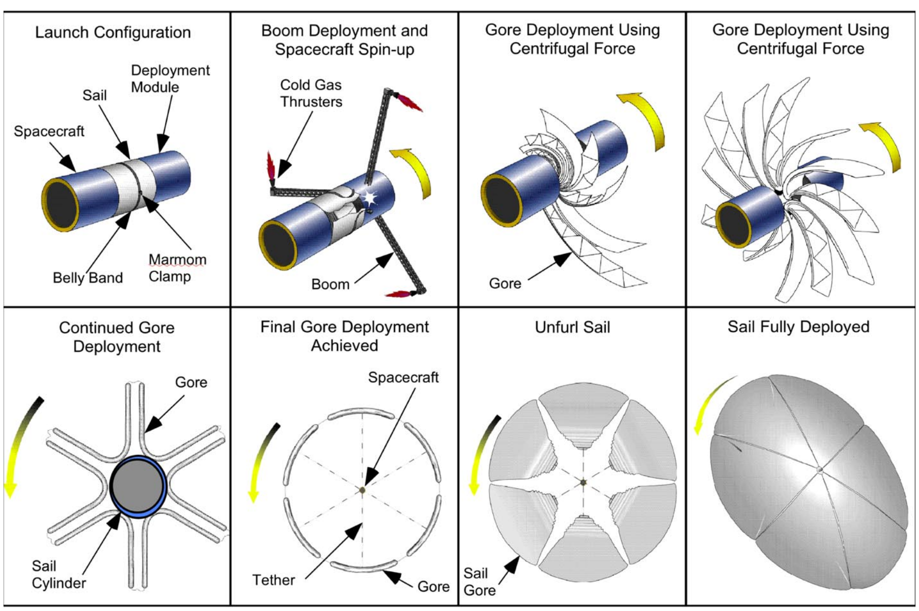

Figure 10. Solar sail deployment sequence

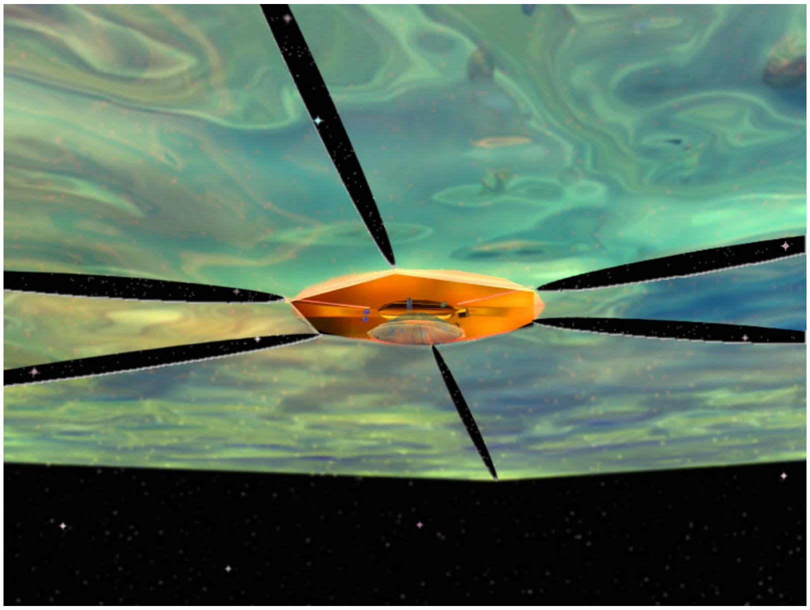

Figure 11. Spacecraft with sail fully deployed (central view)

rail, and the rail, on bearings that decouple the sail-cylinder and spacecraft spin rates, can rotate to any position required for full sailcraft control.

Sun-shade. The Sun-shade is similar in concept to the Next Generation Space Telescope (NGST) Sun-shade. It consists of multilayer kapton, with the kapton sheets set apart by structure. The Sun-shade is inflated and jettisoned as soon as possible after perihelion.

Spacecraft bus with instruments. After reaching 5 AU, the sail is jettisoned and the spacecraft continues toward deep space (Figure 12). During separation, the high inertia of the sail will ensure that it will change attitude very slowly using a small spring-driven separation

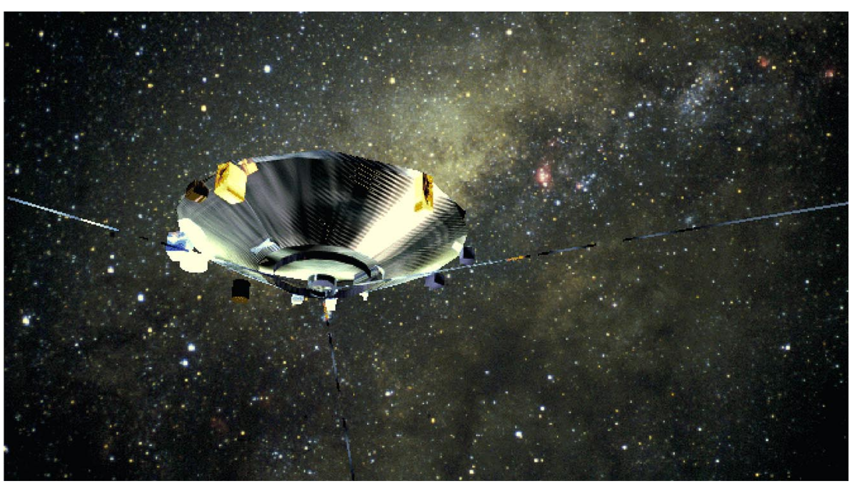

Figure 12. Spacecraft after sail jettison

device, providing easy spacecraft separation. To minimize structure mass, the 2.7-m antenna functions as the main frame of the spacecraft. Subsystems and instruments are mounted on the back side of the antenna, with those requiring a field-of-view past the antenna rim mounted near the perimeter. The antenna is assumed to be a multi-functional structure with all electronics and cabling integrated (Karmon and Chau 1999). Power will be sent to the instruments via optical links and an integrated flex cable on the back of the antenna. The total system cabling mass is estimated to be one-quarter of the value if current technologies were to be used. The structures and mechanisms equipment list and mass table are given in Table 4.

V. Key Technologies. Several elements requiring technology development are incorporated within the baseline design. These key items include the solar sail, telecommunications, power, thermal design, and instrumentation. Table 5 summarizes the technology drivers and those key technology elements required to implement the baseline mission and system concept.

VI. Other Options Considered

Alternate Propulsion Options (Non-Solar Sail)

Chemical propulsion with gravity assist maneuvers. A previous mission study for an interstellar probe was conducted by Mewaldt et al. (1995). The mission scenario examined in that study delivered a 200-kg spacecraft with a 27-kg instrument package to a heliocentric distance of 200 AU in a trip time of 25 years or less. Chemical propulsion with planetary gravity assist maneuvers was chosen for primary mission ΔV.

Table 4: Structures and Mechanisms Equipment List

Component	No. of Flight Units	Mass/Unit (kg)	Total Mass (kg)	Tech. Readiness Level
Primary Structure	1	12.068	12.068	6
Secondary Structure	1	1.448	1.448	6
Instrument Mountings	1	2.5	2.5	6
Spacecraft CG Offset System	3	1.667	5	6
Booms for Instruments	3	1.167	3.5	6
Interface and Integration Heat Shield	1	.145	.145	6
Balance Mass	1	3.618	3.618	6
Adapter (S/C Element)	1	2.202	2.202	6
Cabling	1	5.867	5.867	6

Table 5: Key Technologies and Drivers

Technology Drivers	Key Technologies
Mission Flight Time: 200 AU in 15 yrs	Solar Sail: Areal density of 1 gm/m^2 400-m diameter with control
Data Return: Provide 350 bps at 200 AU	Ka-band phased array with 10-kW Ka-band uplink
Power Required: 106 BOL /8.5 kg	Advanced ARPS
Close Flyby of Sun: Survive 0.25 AU Perihelion Flyby	System approach for thermal loading of ~550 K
Low Mass/Low Power Systems: Instruments, S/C Structure, Packaging, etc.	Instruments: 12 instruments < 25 kg; enhanced science resolution/quality Packaging: Integral design of structure and electronics

Trajectories for gravity assist (GA) using Jupiter (J), the Earth (E), Venus (V), and powered solar flybys (S) were considered in the combinations of JGA, JSGA, EJGA, EJSGA, and VEEJSGA. The conclusion of the study was that an Atlas launch vehicle would support delivery of a 200-kg spacecraft to 200 AU in approximately 25 years. It assumed the spacecraft would have on-board ΔV, the heliocentric velocity would be limited to 14 AU/year, an Earth flyby with radioisotope power supplies would be required, and launch would be no sooner than the first decade of the new millennium.

Nuclear electric propulsion. The evaluation of nuclear electric propulsion systems for a heliopause mission focused on an advanced reactor-based concept with a total power plant specific mass well below 30 kg/kWe. Reactors are essentially nonradioactive at launch. The reactor would be activated at a positive C_3 (beyond Earth escape) to power a krypton-fueled ion propulsion system. The propulsion system would carry the Interstellar Probe science payload on an indirect trajectory (heliocentric spiral trajectory), building up to a final velocity of approximately 25 AU/year after a 10-year run time. Payload mass was studied parametrically from 50 to 250 kg, and trip times to 250 AU ranged from 16 to 20 years. Atlas III, Delta IV, and Proton Launch vehicles were considered.

Alternate Communication Technologies Considered

Optical communication. Optical communication offers a lower mass, lower power, and lower volume (Table 6) telecommunications subsystem than Ka-band. It provides little advantage, however, in meeting the telecommunication requirements while in the inner solar system ($\leq$ 2.2 AU). This is because, during the highly varying Sun-Probe-Earth angles of the inner solar system trajectory, RF LGAs and MGAs are required to support the telecommunications link.

Table 6: Ka-Band and Optical Downlink Comparison

Telecom System	Data Volume (Mbps/week)		Track Duration (hr)	Contingency	Downlink Data Rate (bps)	
	Science	Engineering		Weather	Multiplier	
Ka-Band	15.12	3.02	16	90%**	1.1	350
Optical	15.12	3.02	11*	70%	1.43	655

* Assumes optical ground stations at Hawaii and Goldstone
** Optical communication is more sensitive to cloud cover than RF

Inflatable HGA. A prototype for an inflatable flat-plate array is in the early stages of development. It weighs 13 kg (not including the inflation system, which weighs 4 to 5 kg) and has a demonstrated 55-dB gain at Ka-band. At this time, the antenna is too large for the ISP baselined launch vehicle. It is recommended, however, that future studies continue to examine inflatable antenna options.

VII. CONCLUSIONS. A baseline end-to-end mission/system design, based on advanced technologies expected to be available in the 2007 time period, is described above. A key design feature is the application of an advanced solar sail that is expected to evolve, with flight demonstration(s), over the next seven or so years. The Interstellar Probe mission concept described in this paper will not only advance science knowledge with the first exploration of the boundaries of our solar system with the galaxy, but will also lead technology development for deep space exploration well into the 21st century.

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